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Simulation of Crack Propagation in Engine Rotating Components Under Variable Amplitude Loading

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Abstract

The crack propagation life of tested specimens has been repeatedly shown to strongly depend on the loading history. Overloads and extended stress holds at temperature can either retard or accelerate the crack growth rate. Therefore, to accurately predict the crack propagation life of an actual component, it is essential to approximate the true loading history. In military rotorcraft engine applications, the loading profile (stress amplitudes, temperature, and number of excursions) can vary significantly depending on the type of mission flown. To accurately assess the durability of a fleet of engines, the crack propagation life distribution of a specific component should account for the variability in the missions performed (proportion of missions flown and sequence). In this report, analytical and experimental studies are described that calibrate/validate the crack propagation prediction capability for a disk alloy under variable amplitude loading. A crack closure based model was adopted to analytically predict the load interaction effects. Furthermore, a methodology has been developed to realistically simulate the actual mission mix loading on a fleet of engines over their lifetime. A sequence of missions is randomly selected and the number of repeats of each mission in the sequence is determined assuming a Poisson distributed random variable with a given mean occurrence rate. Multiple realizations of random mission histories are generated in this manner and are used to produce stress, temperature, and time points for fracture mechanics calculations. The result is a cumulative distribution of crack propagation lives for a given, life limiting, component location. This information can be used to determine a safe retirement life or inspection interval for the given location.

Introduction

The U. S. Army is changing its life management philosophy for critical rotating components in rotorcraft turbo-shaft engines. A 'safe-life' approach has been the Army's traditional means of determining component retirement lives. For 'safe-life,' a minimum calculated fatigue crack initiation life dictates when the component is retired from service. Although the 'safe-life' approach tends to be conservative and relatively easy to implement, it also suffers from several shortcomings, i.e.: 1) most components are retired with significant remaining crack initiation life, 2) unforeseen damage, such as scratches incurred during disassembly/assembly, is ignored, and 3) there is no established 'early warning' mechanism to identify changes in use and errors in life prediction.

The U. S. Air Force has pioneered a very different life management philosophy. The Air Force's methodology emphasizes planned inspections to detect damaged components while safely maximizing the use of undamaged components. This approach has often been termed retirement for cause (RFC). The RFC approach is based on fracture mechanics as the means of determining component lives. In RFC, preexisting flaws are assumed to exist at a size just below the detection threshold of the chosen inspection technique. The crack propagation life from this threshold to a fracture critical size is calculated for all high stress locations to determine the life limiting component location. Planned inspection intervals for these high stress locations are set to a fraction of the predicted life of the life limiting location to ensure that all propagating cracks are discovered well before they might cause component failure. The U. S. Army is adopting a tailored version of this approach.

Loading of rotorcraft engines, and in particular military rotorcraft engines, is typically far more variable than in fixed wing aircraft. Army rotorcraft perform a number of different missions with often very different loading histories. This variation in component loading can have a marked influence on the crack propagation life. For example, rapid excursions to high power settings can cause overloads that may retard crack growth rates. Conversely, many small stress excursions at high R-ratios can significantly shorten lives. This additional variability in load history can have a strong influence on the expected life distribution. Given that the Army utility helicopters are used in a variety of missions and the impending implementation of RFC, the effect of this variation on crack propagation life should be quantified.

A methodology is presented herein for predicting the distribution in the crack propagation lives of rotorcraft engine rotating components subjected to realistic mission loading histories. The methodology for determining the mission mix is based on direct Monte Carlo simulation of the engine loading given that the engine will likely be used in multiple aircraft performing a variety of missions. The development of this methodology is predicated upon the ability to accurately model fatigue crack growth under variable amplitude loading. The crack propagation life prediction code, FASTRAN II, was used to perform the life calculations. In order to calibrate and confirm the effectiveness of this code, an experimental program was devised and performed. Results are presented that validate the underlying crack propagation model.

Fatigue Crack Growth Under Variable Amplitude Loading

Models that predict load interactions for cracks propagating under variable amplitude loading, fall into two broad categories. The first group may be classified as overload plastic zone type models, which include the Willenborg [1] and Wheeler [2] models. These models calculate the effective crack driving force as a function of the ratio of the overload plastic zone size to the plastic zone size due to the current load amplitude as well as the location of the crack tip in relation to the overload plastic zone. Models of this type are widely used, especially by the engine companies who have invested significant effort in tailoring these models to their requirements.

The other class may be called crack closure models. They are based upon the original finding by Elber [3] who showed that crack regions behind the crack tip can close prior to reaching the minimum load. The stress intensity range over which the crack tip is open, termed ΔK_{eff}, is assumed to be the true crack driving force. The premature closure of the crack tip region is due to the crack growing through previously plastically deformed material which can come into contact behind the crack tip. The effective stress intensity factor range ($\Delta K_{eff} = K_{max} - K_o$) will therefore be somewhat smaller ($K_o \geq K_{min}$) than that computed without accounting for this effect. A model of this type is used in this work. The effective stress intensity factor range, ΔK_{eff}, is given by:

$$\Delta K_{eff} = Y \cdot (\sigma_{max} - \sigma_0)\sqrt{\pi a} \qquad \text{Eq. 1}$$

where a is the crack depth, Y is a geometric correction factor, σ_{max} is the maximum stress, and σ_o is the stress at crack closure.

The cyclic crack growth rate, assuming a Paris Law, is related to the effective stress intensity range by (Eq. 2):

$$\frac{da}{dN} = C(\Delta K_{eff})^m \qquad \text{Eq. 2}$$

where C and m are material constants. The magnitude of the crack closure stresses (σ_o) is a function of the applied loading history, the crack tip constraint and the size and magnitude of the crack wake field behind the crack tip. A widely recognized crack propagation life prediction code, based upon the crack closure model, is FASTRAN II developed by Newman [4] at the NASA Langley Research Center.

The attractiveness of the crack closure model for engine applications is its ability to accurately predict crack growth of small flaws. Rotating engine components, such as disks, are highly stressed due to their high rotational speed. Because of these conditions, the critical crack size which results in failure is considerably smaller than typically encountered in airframe components. The crack closure models have been shown [5] to reliably predict the early stages of crack development where the crack wake is not yet fully developed and thus the ΔK_{eff} is greater than would be predicted by the overload plastic zone type models. Thus the authors believe that the crack closure models are appropriate where the critical crack sizes are relatively small i.e., lifing of engine rotating components.

The crack propagation analysis program FASTRAN II was used to estimate crack propagation lives in this study. The program contains a library of stress intensity solutions for a variety of crack geometries. The solution chosen for the sample problem examined in this report was a uniaxially loaded surface crack in a prismatic body (Figure 1). The overall dimensions of this prismatic body approximate the dimensions of the appropriate cross section of the second stage gas generator turbine disk in the UH-60 Blackhawk utility helicopter's T700 engine. This solution was also used to perform crack propagation calculations for the K_b bar specimens used in the experimental program. Stress amplitudes and temperatures were taken from calibrated thermal and stress analyses of this component. The failure criterion used by FASTRAN II to discontinue the crack propagation analysis was a calculated stress intensity factor range in excess of 90% of the input material fracture toughness (K_{IC}).

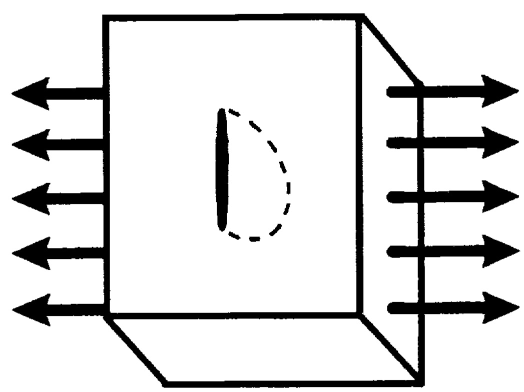


Figure 1: Schematic of prismatic body with surface crack.

The loading histories analyzed in this study were a randomized mix of utility helicopter mission histories. The U. S. Army has specified that the UH-60 Blackhawk helicopter performs ten different missions at three different ambient conditions yielding a total of 30 distinct engine histories (Table I). The purpose of specifying these mission profiles was to enable more realistic fatigue life calculations. It is well known that minor excursions (load ranges less than maximum minus minimum) cause additional damage both in crack initiation and propagation. Table I describes the specified portion of the overall engine life spent in each of these missions.

Table I: Army Mission Mix Specification

		Ambient Conditions			
	Temp.	15 °C	35 °C	35 °C	
	Elevation	Sea Level	Sea Level	1220 m	
Mission					Totals
Troop Assault		5.10	5.80	0.50	11.40
Resupplying		3.30	3.80	0.30	7.40
Aeromed Evacuation		10.50	11.90	0.90	23.30
Replacing Units		3.00	3.40	0.30	6.70
Transport of Recon.		2.30	2.70	0.20	5.20
Reinforce/Reposition		4.70	5.40	0.40	10.50
Troop Extraction		2.60	3.00	0.20	5.80
Aerial Command Post		2.00	2.20	0.20	4.40
Sling Load		3.80	4.30	0.30	8.40
Training		7.60	8.60	0.70	16.90
Totals		44.90	51.10	4.00	100.0

All values in % of engine operating life

Calibration and Validation of Crack Growth Algorithm

To validate the accuracy of the FASTRAN II crack propagation analysis code for variable amplitude loading of a turbine disk alloy at elevated temperature, an experimental program (10 specimens) was performed. The purpose of the experimental program was twofold. The first was to generate relevant baseline data necessary to calibrate the crack closure model used by the FASTRAN II program. The second was to generate variable amplitude loading experimental data that could be directly compared with FASTRAN II predictions.

Mechanical Testing Procedure

All specimens were cut from a single retired René 95 disk (Fig. 2). René 95 is a powder metallurgy, nickel-base superalloy that is used for most of the gas generator turbine rotating components in the UH-60 Blackhawk's T700 engine. The experiments were performed at 400 °C to more closely simulate the engine operating environment at the fracture critical locations: the disk bore and bolt/cooling air holes.

Fatigue crack growth testing was performed with an axially loaded surface cracked specimen geometry, the K_b bar (Fig. 3). The K_b bar geometry was thought to be more representative of the actual flaws and high applied stresses expected in turbine disk applications than the more conventional compact tension specimen geometry. The gage section dimensions of the K_b bar specimen were nominally 10.2 mm wide by 4.3 mm thick. Crack lengths were monitored with a direct-current electric-potential (DCEP) technique. DCEP in this program was calibrated using marking cycles, heat tinting, and post failure crack length measurements. Testing was performed on a computer controlled, servo-hydraulic, test frame equipped with a clamshell furnace.

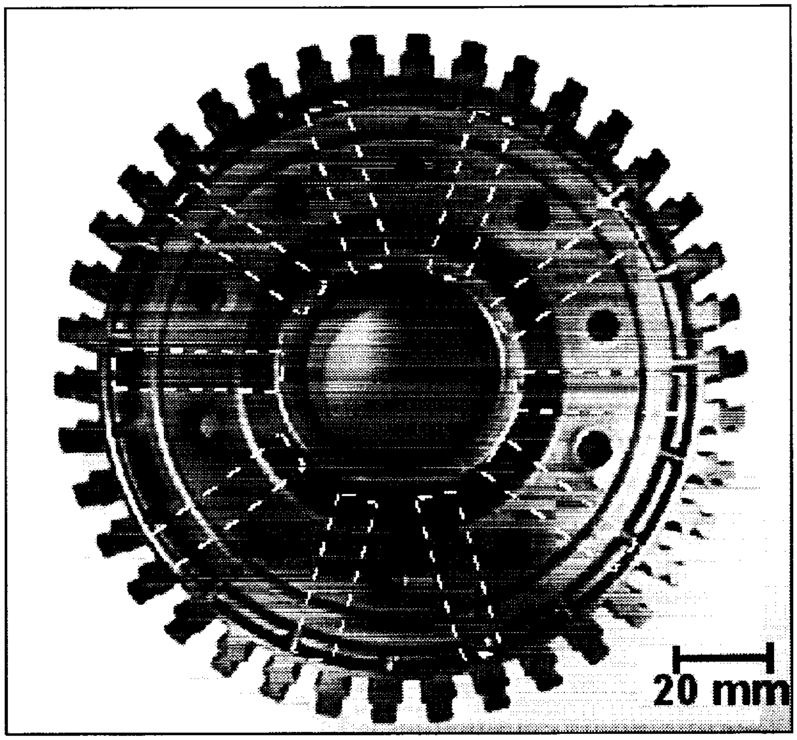


Figure 2: Fatigue crack growth specimen blank locations; T700 Stage 2 gas generator turbine disk.

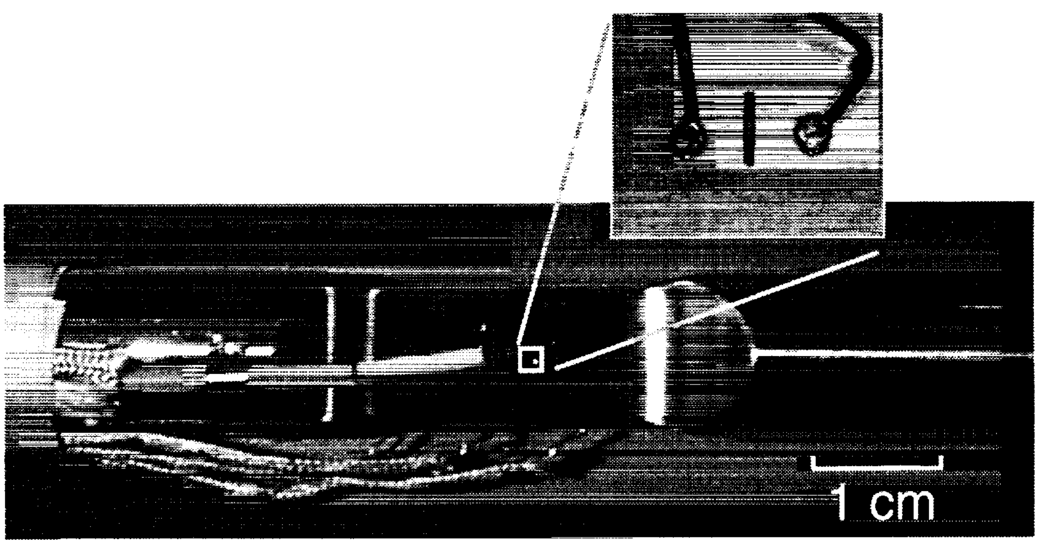


Figure 3: Kb Bar specimen instrumented for elevated temperature crack propagation experiment. Inset is a close up of an EDM notch with spot welded DCEP probes.

The cracks in the specimens were not allowed to propagate through the specimen thickness. The experiments were stopped at a crack depth where the accuracy of the analytical solution for the DCEP calibration was expected to degrade (approximately 50-60% through thickness).

Calibration Experiments

To properly calibrate the crack closure model, constant amplitude fatigue crack growth experiments were performed at three different R ratios ($R = \sigma_{min}/\sigma_{max}$): 0.05, 0.5 and 0.7. In Figure 4, the experimental data from these calibration experiments is displayed as a function of the applied crack driving force, ΔK_{app}. Figure 5 shows the same data set corrected for crack closure with Newman's formulation for the effective crack driving force, ΔK_{eff}. This formulation is used by the FASTRAN II software to estimate the effect of crack closure on the crack propagation rate. The stress intensity geometric correction formula for the K_b bar specimen was taken from Newman and Raju [6].

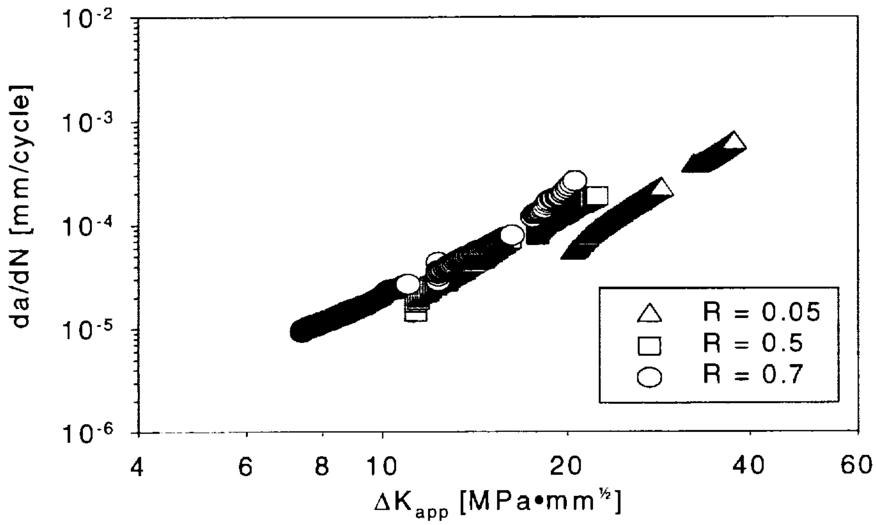


Figure 4: da/dN vs. ΔK_{app} for René 95 at 4008C.

As seen in Figure 4, the R=0.05 data is considerably different than the 0.5 and 0.7 data, showing slower crack growth rates for the same ΔK_{app}. This difference is expected since at a 0.05 R-ratio, the crack closes at a higher applied stress than the minimum stress. But when the stress intensity factor range is corrected for the early closure of the crack faces ($\Delta K_{eff} = (K_{max} - K_{closure}) < (K_{max} - K_{min})$) using Newman's formulation [4], there is a better correlation between the three different R-ratios, (see Fig. 5). Hence, the ΔK_{eff} parameter plotted in Figure 5 represents a true measure of the actual crack driving force, independent of the applied load ranges. The curve fit of the crack growth rate versus the effective stress intensity factor range, determined from these three constant amplitude experiments, is supplied to FASTRAN II for use in predicting the variable amplitude loading experiments.

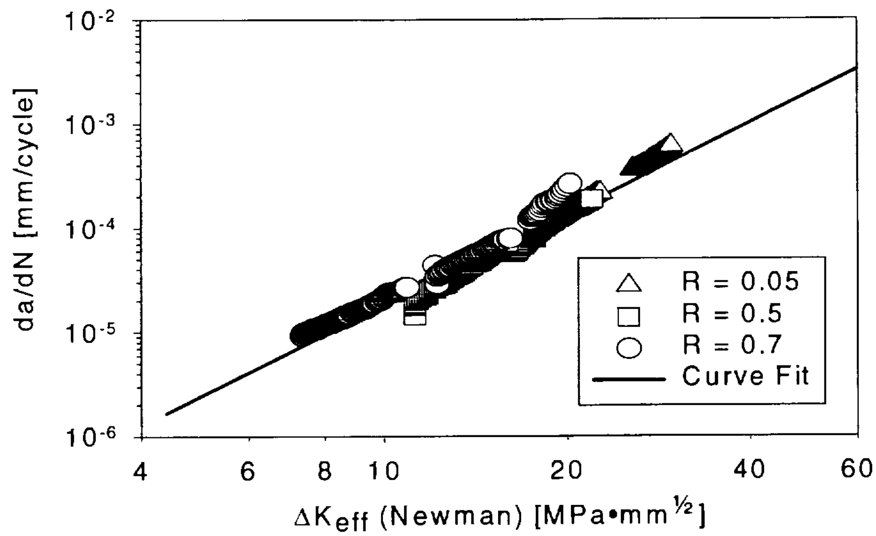


Figure 5: da/dN vs. ΔK_{eff} for René 95 at 4008C.

Validation Experiments
The variable amplitude loading experimental program, consisted of 5 experiments. Three different repeating load sequences were applied to K_b bar test specimens at 400 °C. Load sequence 1 (LS 1) (Fig. 6a) closely resembles the Army specified training mission. Load sequence 2 (LS 2) (Fig. 6b) is essentially the same as sequence 1 with a few additional large excursions (each roughly equivalent to a take-off - ground idle excursion). Load sequence 3 (LS 3) (Fig. 6c) was run to test the extremes of the crack closure model's predictive capability. It includes excursions covering various ranges of loading R ratio.

The FASTRAN II analysis of the three loading spectra showed excellent agreement with the experimental results. Comparison of the experimentally observed crack depth versus the number of mission spectra repeats applied with the FASTRAN II predictions for each of the three spectra are shown in Figure 7. Only the data from one experiment for each loading spectrum is displayed. The predicted crack depths faithfully replicate those observed in the experiments. The predicted final crack depths were all within a factor of 1.5 of the experimentally observed values (Fig. 8).

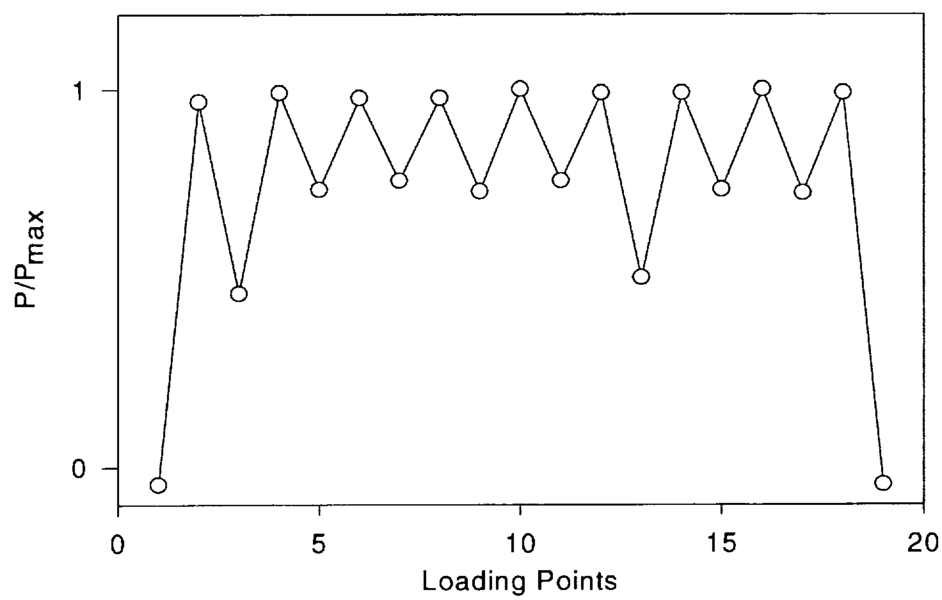


Figure 6a: Experimental Loading History 1 (LS 1).

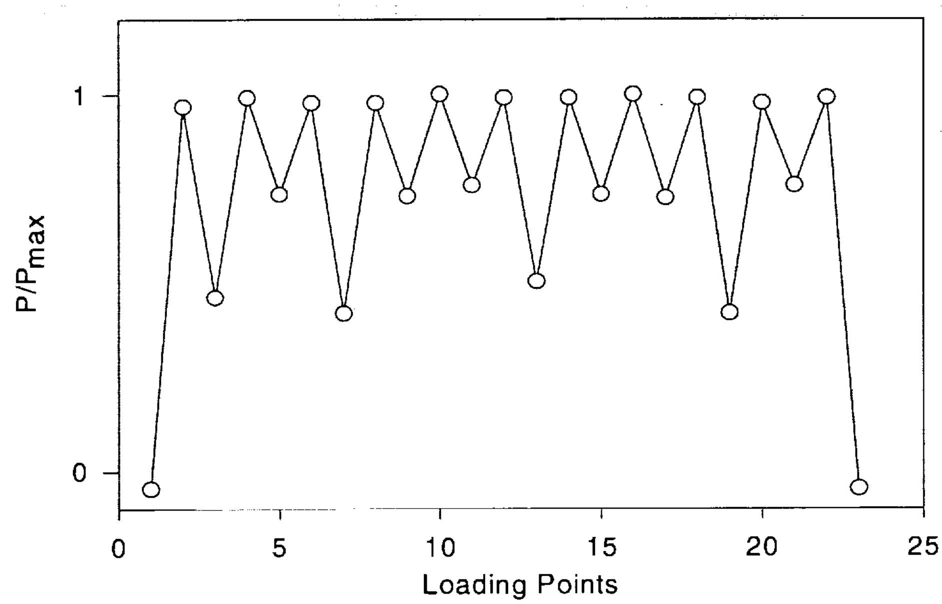


Figure 6b: Experimental Loading History 2 (LS 2).

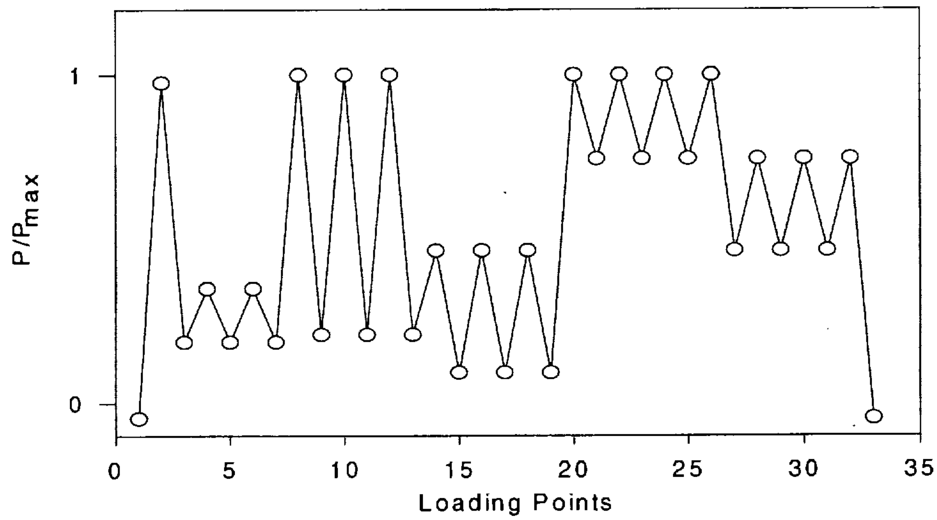


Figure 6c: Experimental Loading History 3 (LS 3).

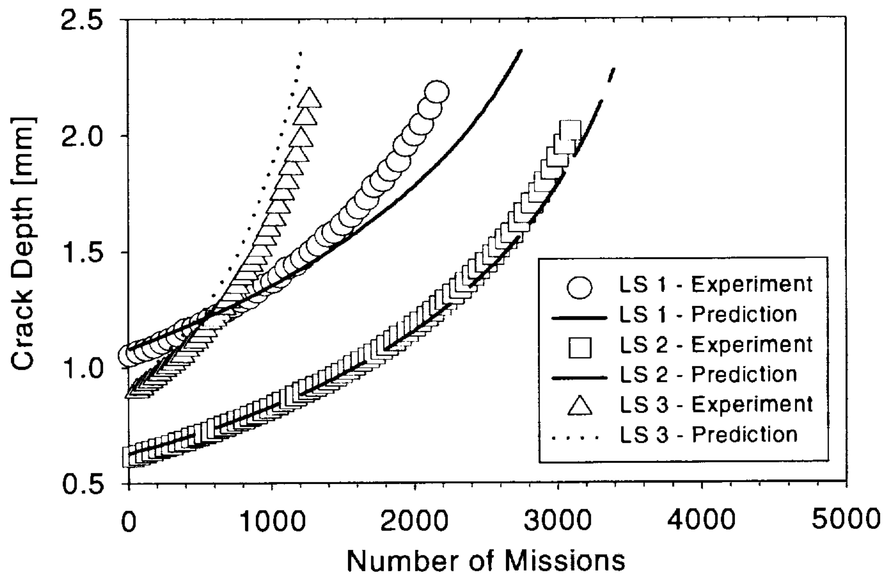


Figure 7: Correlation of FASTRAN II mission mix predictions with experimental crack depth data.

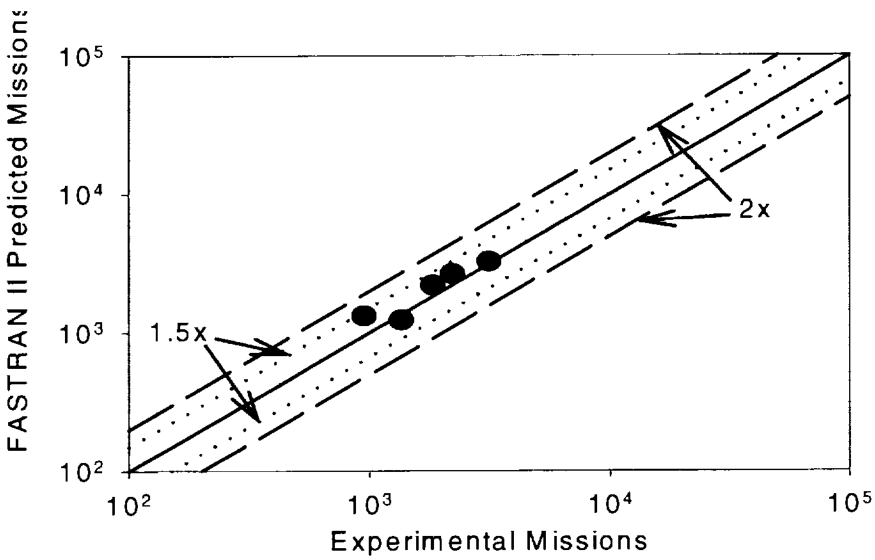


Figure 8: Experimental vs. predicted crack propagation lives for block loaded K_b bar FCG specimens.

In order to evaluate/demonstrate the need to correct the crack driving force for load interactions, the crack growth data were modeled using a Miner's type (linear) summation of all the different cyclic segments without regard for load interactions. The crack growth rates for a given load excursion, at a given R ratio, were calculated using Walker's methodology [7] for evaluating the influence of the load ratio. The Walker model uses ΔK_{app} as the crack driving force. The results of the analysis, performed on the LS 3 spectrum (Fig. 6c) are presented in Figure 9. As shown, the crack growth rate calculated using this linear summation approach is substantially faster than was experimentally measured, amply demonstrating the presence of load interaction effects.

Also, an analysis of the minor cycles' influence on the crack growth behavior was performed. This analysis consisted of a FASTRAN II run with a single load excursion from the minimum stress to the maximum stress and back down to the minimum stress (all the minor cycles removed from the load history). As shown in Figure 9, the major excursion in the load histories could only account for a small fraction of the observed crack growth. Thus, the minor cycles' contribution to crack growth, for this particular spectrum, is significant.

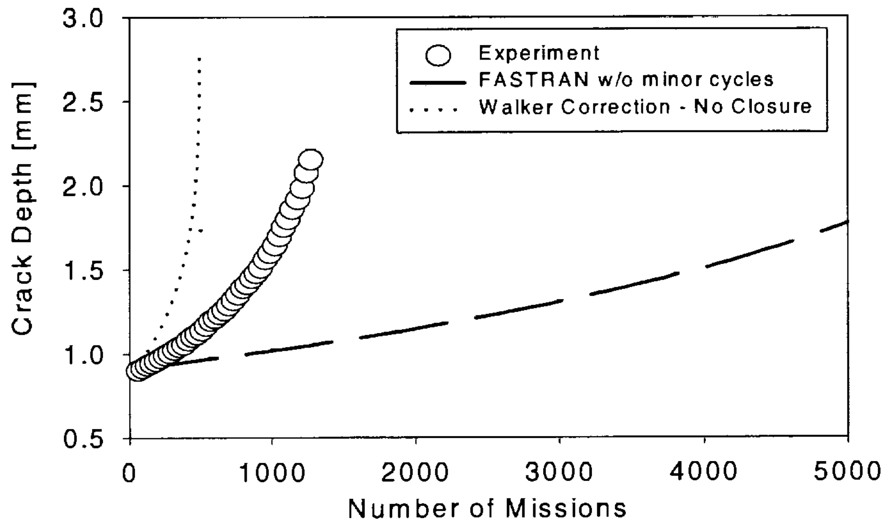


Figure 9: Influence of crack closure and minor cycles on the predicted crack propagation rate. The LS 3 spectrum was used in displayed results.

Simulation of Mission Mix Histories

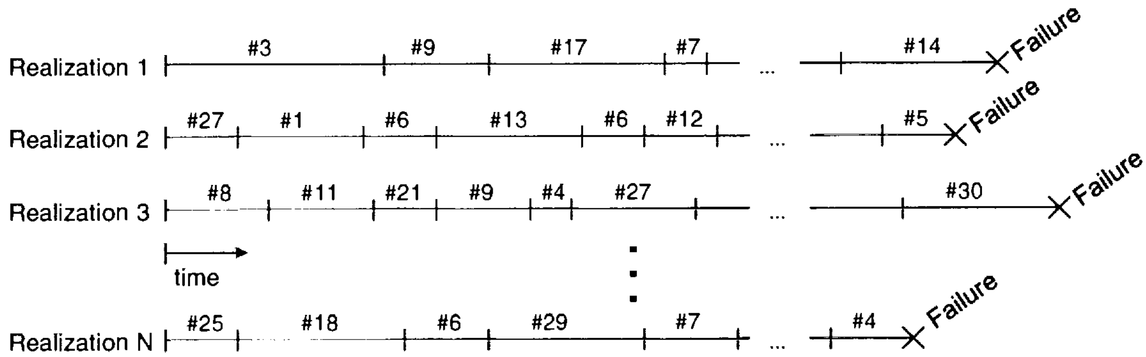
Given that the FASTRAN II crack propagation analysis code proved successful in predicting the crack growth behavior for the various spectrum loading sequences, it was deemed an appropriate tool for the study of mission mix history on the overall fatigue crack growth life.

There are several ways to interpret the Army's mission mix specification. The most literal interpretation is that each engine will experience the exact mission fractions and ambient conditions specified in the mission mix. In practice, this is highly improbable. It is far more likely that the mission mix is an estimate of the fleet usage. This would mean that an appropriate fraction of the fleet would experience the specified missions, for example, 0.5% of the fleet flying time (given a fleet of 2000 aircraft, one aircraft might fly this mission continuously or two aircraft half the time, etc.) will be used for the troop assault mission on a 35 °C day from a base at 1220 m. Therefore, the more severe missions (as well as the more benign missions) are not spread evenly to all engines. This leads to a larger variance in the distribution of the expected crack propagation lives. In addition, the longer the interval between changes in mission for the engines, the larger the spread in the fatigue lives.

In describing this simulation technique several terms will be used that require unambiguous definition: A mission is a loading sequence that approximates the loading imposed on a component during a single rotorcraft flight, from engine start-up, through takeoff and flight maneuvers, to landing and engine shut-down. A block is defined as a number of sequentially repeated missions of the same type (e.g., any number of repeated Troop Assault missions all at sea level standard conditions would be considered a block). A block of missions is intended to simulate the repeated use of an engine for a specific mission while stationed with a unit. The block length is defined as simply the number of repeated missions in a block. A realization is defined as a sequence of blocks that terminates with component failure. A realization can be thought of as the simulated loading history of a single engine.

To determine the crack propagation life of a component, all potentially life limiting (high stress) locations are analyzed to predict the crack growth life from a flaw of a known size (as determined by the resolution and reliability of the inspection technique). The feature chosen for this crack propagation simulation was the bore of the second stage gas generator turbine disk. A crack propagating from this location would eventually lead to a catastrophic uncontained engine failure. To estimate the effect of the random mission

mix loading, a direct Monte Carlo simulation technique was formulated. This method involved generating a number of blocks of mission loading sequences that were then fed into the FASTRAN II crack propagation analysis code (Fig. 10). This was repeated (16-64 realizations) with new randomly generated sequences of mission blocks to get a distribution of crack propagation lives.



#1 Troop Assault SLS	#11 Troop Assault 35°C	#21 Troop Assault 35°C HA
#2 Resupply SLS	#12 Resupply 35°C	#22 Resupply 35°C HA
#3 Aeromed. Evac. SLS	#13 Aeromed. Evac. 35°C	#23 Aeromed. Evac. 35°C HA
#4 Replacing Units SLS	#14 Replacing Units 35°C	#24 Replacing Units 35°C HA
#5 Trans. of Recon. SLS	#15 Trans. of Recon. 35°C	#25 Trans. of Recon. 35°C HA
#6 Reinforce/Repos. SLS	#16 Reinforce/Repos. 35°C	#26 Reinforce/Repos. 35°C HA
#7 Troop Extract. SLS	#17 Troop Extract. 35°C	#27 Troop Extract. 35°C HA
#8 Aerial Comm. Post SLS	#18 Aerial Comm. Post 35°C	#28 Aerial Comm. Post 35°C HA
#9 Sling Load SLS	#19 Sling Load 35°C	#29 Sling Load 35°C HA
#10 Training SLS	#20 Training 35°C	#30 Training 35°C HA

Figure 10: Schematic of random mission block loading. Note both the random selection of missions and the variation in the block length.

Several assumptions have been made to simplify the FASTRAN II simulation and to fill in for certain unknowns about the actual usage of the engine. First, it was assumed that the engines are 'pooled' at the depot. By this it is meant that an engine returned to the depot for maintenance will not necessarily return to the original 'owner' but will go to the first unit requiring an engine. This allows us to model the selection of each block of missions with an appropriately partitioned uniform random variable. The probability of selecting a particular mission and ambient condition is directly linked to the fraction of time the engine is expected to spend performing that mission/ambient condition combination as described by the Army mission mix specification (Table I).

Another, and perhaps oversimplifying, assumption is that, while stationed with a unit, the aircraft will perform one mission exclusively. For example, aircraft stationed at a training facility perform only the training mission. It was also assumed that the block length (number of missions between removal of the engines for overhaul at the depot) could be modeled as a Poisson process. The purpose of using a Poisson distributed random variable, as opposed to a fixed block length, was to more closely model the actual engine usage (the T700 has no scheduled overhaul interval). A Poisson process is defined as a discrete process where the intervals between events are exponentially distributed. The probability distribution is defined by (Eq. 3):

$$P(x) = \frac{\lambda^x}{x!} e^{-\lambda} \qquad \text{Eq. 3}$$

where P(x) is the probability that x events occur and λ is the mean number of events. This equation may be solved for the number of events, x, and may be used to generate a Poisson distributed random variable. The Poisson distribution parameter, λ, for the purposes of this simulation, is equal to the mean block length and the random variable, x, will be defined as the Poisson distributed block length. The method of generating a Poisson distributed random variable was taken from Press et al. [8].

The simulation method was used to execute six runs of 16 realizations each using various values for the mean block length, λ (Table II). An additional run of 64 realizations with λ = 165 missions, was performed as a measure of the sample to sample variability. The first row of Table II, labeled λ = 1, was actually run with the block length fixed to one mission. This is roughly analogous to running the 30 missions, in proportion to the mission mix specification, on each of the engines.

Table II: Simulation Results

λ [missions /block]	Realizations	Mean [hrs]	Standard Deviation [hrs]	-3σ Life [hrs]
1	16	8496	182.1	7950
33	16	8522	315.3	7576
66	16	8520	628.4	6472
99	16	8662	715.0	6517
132	16	8607	1176.1	5078
165	16	8578	1344.0	4546
165	64	8476	772.1	6160

The increase in the dispersion of the simulated crack propagation lives as the expected number of repeated missions between change of mission increases is clearly shown in Figure 11.

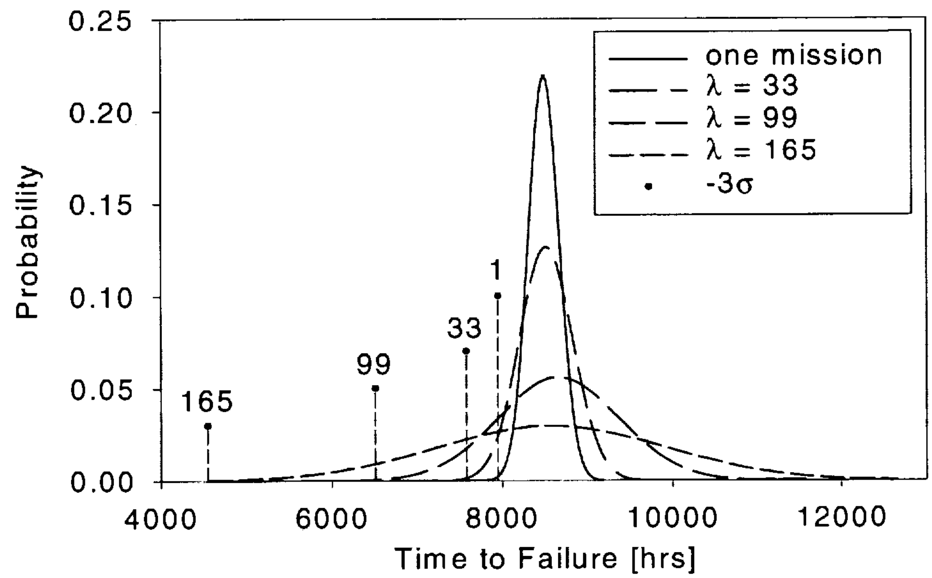


Figure 11: Normal distributions derived from simulation results. The labeled drop lines indicate life at mean minus 3 standard deviations for each distribution.

The last row of Table II shows the results of a repeat of the λ = 165 mission simulation with 64 realizations. The standard deviation in this sample was considerably smaller than in the 16 realization simulation run at the λ = 165 missions. The reason for this difference in the variance cannot be readily explained. However, it should be noted that the variance is likely to fluctuate from one sample to the next, especially when the sample size is relatively small. The difference in the simulated standard deviations may simply illustrate the sample to sample variation in the variance and may indicate that 16 realizations of the randomized mission loading sequence is inadequate to provide a reasonable estimate of the underling population's standard deviation. In spite of this discrepancy the trend of increasing variance with simulated block length seems to hold true.

Discussion

If one ignores the loading interaction that may occur in changing missions (variation in the maximum and minimum stresses from one type of mission to the next is small when compared to the variations in the loads within each mission), a linear summation approach might be used to calculate the mission mixed loading crack propagation life distribution. Given the calculated crack propagation lives for each of the missions, an analytical distribution function for the probability of failure can be derived if the following question can be answered: What fraction of the possible combinations of mission blocks, of Poisson distributed length, lead to failure in a given time, t? Numerical integration of the probability density function is possible but the number of combinations is daunting. For example: given a Poisson distribution parameter for the block length, λ, of 165 missions and assuming an average mission length of 1.46 hours per mission, at 3675 hours of component operation there would be approximately 15 blocks of missions flown, each block with one of the 30 possible missions. Assuming no distribution in the fatigue lives of the individual missions, i.e., each mission contributes exactly 1/(total mission cycles to failure for a specific mission) fractional damage to the component and assuming that each block is equally possible and independent, there can be 30^{15} or 1.4×10^{22} possible combinations. In calculating the probability of failure at longer operating times, the number of mission blocks increase and the number of possible combinations grows exponentially. With the additional complications that the missions are not equally probable, the length of the individual missions are not constant, and the number of missions per block is also variable, this becomes an increasingly difficult problem to solve in closed form. If it is assumed that the loading interaction that can occur in changing missions has a marked effect on fatigue crack growth, a closed form analytical solution does not seem feasible.

The results of this study indicate a strong dependence of the crack propagation life on the mission sequence and particularly on the mean block length (expected number of mission repeats between engine removal for overhaul). The expected crack propagation life variance caused by the mission mix loading will necessitate a reduction in the estimated lower bound crack propagation life. If not accounted for in the current component life estimates, retirement lives and/or inspection intervals will need to be shortened.

A basic assumption driving this simulation technique is that the mission sequence has strong effect on crack propagation rate. If it can be demonstrated that the effect of the mission sequence on the crack propagation rate is negligible, or that the mission loading in the prescribed mission mix is not conducive to significant 'mission to mission' load interactions, an analytical solution or a simplified (less computationally intensive) simulation technique might be utilized.

The intrinsic (material dependant) variability in the crack propagation life could also be incorporated into the simulation. This would require an additional random variable to account for the experimentally observed scatter in the crack propagation rate.

Given that the Army specified mission mix is an estimate of the actual usage, that the fleet usage is likely to evolve with time, and that there can be significant pilot to pilot variation in how the aircraft are flown, the contribution to the variability in crack propagation due to these loading variations can potentially be large. Accounting for this loading variability could lead to unacceptably short inspection intervals, increased inspection costs, and a large impact on fleet readiness. To mitigate the loading history uncertainty, a strong argument can be made for onboard monitoring of the aircraft usage. This could allow for a 'real-time' calculation of the available crack propagation life. A more aggressively used engine could be pulled for inspection or retirement, even though the original life estimates (based on the expected usage) would indicate that it was still safe to operate. Conversely, a lightly used engine might not be removed as often for costly inspection/overhaul or outright replacement.

Summary and Conclusions

The FASTRAN II crack propagation analysis code, which is based on the crack closure model, accurately predicts crack propagation in surface cracked K_b bar specimens under variable amplitude loading.

A method of estimating crack propagation life distribution of helicopter engine rotating components subjected to thirty different missions of varying severity, length, and probability of occurrence, has been developed. This method utilizes several random variables to generate simulated engine histories that are then fed into the FASTRAN II crack propagation analysis code.

There is a direct correlation between the mean number of missions between removal for inspection/overhaul and the variance in the predicted crack propagation lifetimes. Extra caution is therefore necessary when determining the retirement life or a safe inspection interval for critical rotating components. A more accurate accounting of the actual usage of the engines would also seem prudent. This might include onboard monitoring of engine usage.

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The crack propagation life of tested specimens has been repeatedly shown to strongly depend on the loading history. Overloads and extended stress holds at temperature can either retard or accelerate the crack growth rate. Therefore, to accurately predict the crack propagation life of an actual component, it is essential to approximate the true loading history. In military rotorcraft engine applications, the loading profile (stress amplitudes, temperature, and number of excursions) can vary significantly depending on the type of mission flown. To accurately assess the durability of a fleet of engines, the crack propagation life distribution of a specific component should account for the variability in the missions performed (proportion of missions flown and sequence). In this report, analytical and experimental studies are described that calibrate/validate the crack propagation prediction capability for a disk alloy under variable amplitude loading. A crack closure based model was adopted to analytically predict the load interaction effects. Furthermore, a methodology has been developed to realistically simulate the actual mission mix loading on a fleet of engines over their lifetime. A sequence of missions is randomly selected and the number of repeats of each mission in the sequence is determined assuming a Poisson distributed random variable with a given mean occurrence rate. Multiple realizations of random mission histories are generated in this manner and are used to produce stress, temperature, and time points for fracture mechanics calculations. The result is a cumulative distribution of crack propagation lives for a given, life limiting, component location. This information can be used to determine a safe retirement life or inspection interval for the given location.

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